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SIMULATION STUDIES FOR PLANNING AN IN-FLIGHT EXPERIMENT TO DEFINE MANUAL GUIDANCE AND CONTROL TECHNIQUES FOR LARGE LAUNCH VEHICLES

by DeLamar M. Watson, Gordon H. Hardy, and Glenn W. Stinnett

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Moffett Field, Calif.

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SUMMARY

The simulator program described in this report was undertaken to define an in-flight experiment to evaluate manual control system concepts for Saturn class launch vehicles. The techniques and manual control system configurations developed in earlier studies were adapted to the uprated Saturn I, which was the vehicle chosen for the experiment. This report discusses the vehicle and manual control system configuration, the handling qualities, and two manual guidance display schemes. An assessment is made of the impact of the proposed experiment on mission reliability.

INTRODUCTION

The astronauts have demonstrated their ability to contribute to mission success by participating in the guidance and control of their spacecraft. During the Mercury and Gemini programs, they have shown a capability to manually control vehicle attitude, to conduct rendezvous maneuvers, to make orbit changes, and to initiate and control reentry. However, during the launch segment of flight, the crews have not had an active manual role, rather, they have been primarily systems monitors.

In a joint effort by Marshall Space Flight Center (MSFC) and Ames Research Center (ARC), piloted flight simulators have been used to evaluate manual control systems for Saturn class vehicles. The feasibility of a rate-augmented manual control system for the S-IC stage of the Saturn V was demonstrated (ref. 1). A follow-on study showed that manual guidance to a nominal trajectory is feasible for the S-II and S-IVB stages (ref. 2). An extensive study was made of the mission reliability contribution a pilot could make by taking over control of the vehicle in the event of a failure in the automatic system (ref. 3). The results of this research indicated that a pilot, using a simple parallel loop, could cope with a variety of launch vehicle subsystem failures and that mission reliability was substantially increased. At the request of the Manned Spacecraft Center, a manual backup scheme based on existing Apollo hardware was evaluated as a potential alternate to the Saturn V iterative guidance system (ref. 4). All of these studies pointed to the feasibility of manual guidance and control of large launch vehicles.

As the feasibility of a manual mode in the control system of Saturn class vehicles became more firmly established, it seemed desirable to carry the program into an in-flight experiment phase. Accordingly, an experiment definition program was initiated for a specific vehicle. The uprated Saturn I was chosen for the study since this was thought to be the first launch vehicle that would be available for an experiment. The experiment definition program was centered on the simulation study described in this report.

The first of four objectives of the simulation study was to adapt to the Saturn I the feedback configuration and filters, the displays, and the pilot procedures developed in the studies reported in references 1 to 3. The second objective was to define maneuvers that would provide maximum information return from the limited launch vehicle flight time. The third objective was to establish baseline data for comparison with flight data. A final objective was to establish that the addition of a manual control system to the Saturn I vehicle would not lower the probability of mission success.

This report contains all the phases of mission planning that were conducted as part of the simulation study. The first section of this report provides a framework for experiment modifications by describing the uprated Saturn I and its flight environment. Section two is devoted to a description of the simulation. Section three contains a discussion of the studies carried out to determine the handling qualities for each of the stages. Guidance studies during the first-stage flight are treated in the fourth section. The reliability of the manually controlled Saturn I is assessed in the concluding section.

SYSTEM DESCRIPTION

Vehicle Description

The uprated Saturn I is a two-stage launch vehicle used to place the Apollo command and service modules into an earth orbit. The major parts of the vehicle are indicated in figure 1, from bottom to top: S-IB first stage, S-IVB second stage, Instrument Unit (IU), and the Apollo payload.

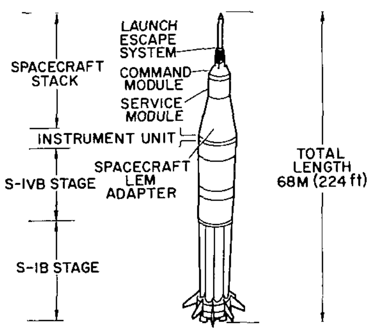

Figure 1.- Saturn I vehicle configuration.

The essential features of the S-IB stage are nine slim propellant tanks and eight engines that each produce 200,000 pounds of thrust. The four inboard engines are fixed. Each of the four outboard engines are swiveled in pitch and yaw by hydraulic actuators to provide control torques. Pitch and yaw motions are imparted to the vehicle by appropriately swiveling all four engines. Roll motions are produced by swiveling the engines differentially. Each propellant tank is

baffled to prevent sloshing. Because the complete Saturn I is long and slim, structural bending is a significant consideration for first-stage flight.

The S-IVB stage has a single engine. Propellant is contained in two tanks - liquid hydrogen in the forward tank, liquid oxygen in the aft tank. Because of the shape of the liquid oxygen tank (two hemispherical segments joined together), sloshing is a significant consideration during second-stage flight. Sloshing in the second-stage oxygen tank is not significant during the first stage of flight because of baffle rings located around the top of the tank. Structural bending is not a serious consideration in designing a control system for second-stage flight because the S-IVB is shorter and stiffer than the Saturn I and operates above the atmosphere. Hydraulic actuators swivel the engine to produce pitch and yaw control torques. Roll control is maintained with the auxiliary propulsion system reaction jets.

An Instrument Unit (IU) is located just ahead of the S-IVB stage. The IU contains a guidance computer, a data adapter, rate and lateral acceleration sensors, and a control computer that produces engine actuator command signals.

The Apollo payload is a modular stack consisting of a Lunar Excursion Module adapter, a service module, a command module, and a launch escape system. The Apollo command module houses a three-man crew. Although the guidance and control system in the Apollo is primarily intended for postlaunch tasks, it can be used to monitor the launch vehicle flight.

Wind Environment

The wind environment is the primary external disturbance during the boost flight and is a major problem for the design of the Saturn first-stage vehicle control system. Two synthetic wind magnitude profiles were used for this study. These profiles (fig. 2) are based on statistical analysis of wind measurements taken at the Air Force Eastern Test Range, Cape Kennedy Launch Area. The two profiles, as indicated, have steady-state values that will not be exceeded 95 and 50 percent of the time, respectively, during the most windy month of the year at Kennedy Spaceflight Center. The accompanying vertical wind shear is not exceeded 99 percent of the same time period for both profiles. The peak value of wind shear occurs near the altitude corresponding to vehicle maximum dynamic pressure. A preliminary investigation showed that the small amplitude gusts discussed in reference 1 had little effect on the manual control problem. Two wind directions were found to have the greatest effect (ref. 1): 135° and 225° relative to the vehicle launch heading.

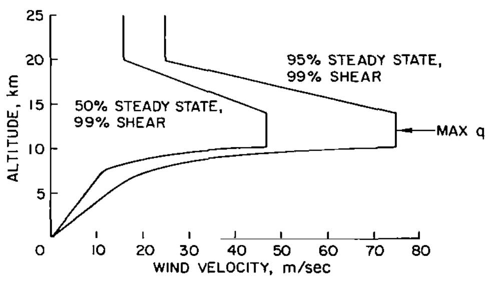

Figure 2.- Wind profiles.

Saturn I Guidance and Control System

The guidance and control system that provides trajectory and attitude control from lift-off to payload separation is located in the Instrument Unit. Guidance command signals are provided by the Launch Vehicle Digital Computer shown in figure 3. During first-stage burn, the vehicle follows a pre-programmed open-loop tilt program to minimize aerodynamic loads during the high dynamic pressure portion of flight. During second-stage burn, the computer iteratively calculates command signals for a "minimum propellant" trajectory to previously stored aim conditions.

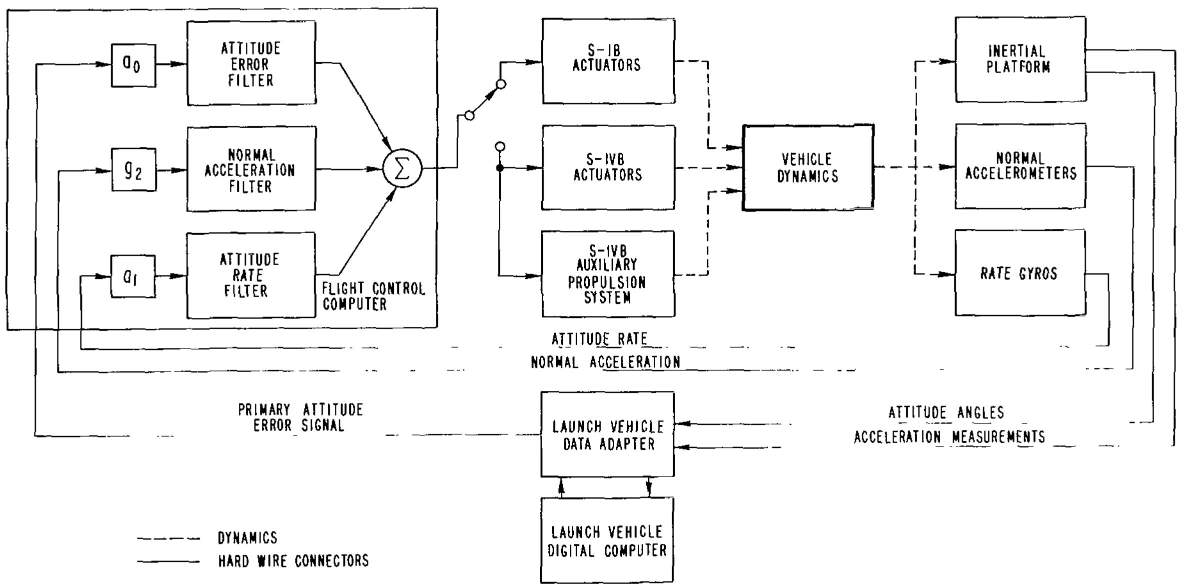

Figure 3.- Block diagram of Saturn control system.

During the Saturn I flight, the control law for the engine deflection angle, β_c, is:

$$\beta_c = a_o N_o \Delta\phi + a_1 N_1 \dot{\phi} + g_2 N_2 \ddot{Y} \qquad (1)$$

where

a_o attitude error feedback gain, deg/deg

a_1 attitude rate feedback gain, deg/deg/sec

g_2 normal acceleration feedback gain, deg/m/sec^2

N_o attitude error filter

N_1 attitude rate filter

N_2 normal acceleration filter

$\ddot{\gamma}$ normal acceleration, m/sec^2

$\Delta\phi$ attitude error angle, deg

$\dot{\phi}$ attitude angle rate, deg/sec

Figure 3 indicates how this control law is implemented in the Saturn launch vehicles. An inertial platform located in the Instrument Unit provides attitude angles and acceleration measurements that are used by the launch vehicle digital computer to calculate an analog attitude error signal. This error signal and signals from rate gyros and normal accelerometers are fed into an analog Flight Control Computer. Each of the signals passes through the filters and gains shown in the figure and are then summed according to equation (1). The feedback gain variables, a_0, a_1, and g_2, are not constant throughout the flight. They are adjusted to compensate for changing vehicle characteristics. During the Saturn I flight, a_0 and a_1 are reduced in three steps at 10-second intervals starting at 100 seconds after lift-off. The normal acceleration gain, g_2, varies as indicated in figure 4. The normal acceleration is used only during the portion of the flight where high dynamic pressure is encountered.

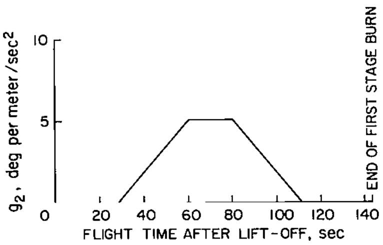

Figure 4.- Accelerometer feedback gain variation.

The control computer on the Saturn I vehicle contains the passive networks mentioned earlier. The attitude error signal is filtered by a first-order time lag. The attitude rate signal passes through a second-order over fourth-order network designed to avoid excitation of the bending modes. Two time lags in tandem filter the accelerometer signal.

Flight Event Sequence

An event sequence for the reference vehicle used for the simulation is shown in table I. This table introduces the three portions of flight that are significant for the flight-test program. The first part is the high dynamic pressure region of flight (high q) that starts near 60 seconds, reaches a peak near 70 seconds, and is essentially over by 100 seconds. As indicated during the wind profile description, the major control problem occurs during the high q part of the flight. The second portion of the flight is from 100 to 140 seconds and is suitable for doing altitude control and guidance display evaluation maneuvers. The feedback gain changes indicated in the table do not interfere with the maneuvers. The third part of flight occurs from 185 seconds, when the launch escape tower is jettisoned, to 350 seconds, when the attitude rate feedback gain reduction takes place. Perturbations which might be introduced by maneuvers done early in the second-stage flight can be eliminated by the iterative guidance system well in advance of orbit injection.

SIMULATION DESCRIPTION

To subject the pilot to all the problem areas he may encounter during the actual flight, it was necessary to construct a flight simulation of the Saturn I-Apollo vehicle that could duplicate both normal and emergency situations. The elements simulated were the Saturn I vehicle dynamics and control system as well as the Apollo pilot's display and hand-rotational controller. The details of the Saturn I simulation given below are followed by a description of the fixed-base cab used to represent the Apollo command module.

Saturn I Vehicle Simulation Model

Separate simulations were used to model each of the two stages of the uprated Saturn I launch vehicle. The separate simulations eliminated the complexity that would be associated with the mechanization of staging operations. The two sets of linearized equations used for the study represent the rotational and translational perturbations of the vehicle relative to a nominal trajectory. The details pertaining to the weight, thrust levels, flight-path programming, etc., were obtained from MSFC.

The equations used for the first simulation included terms corresponding to both normal and emergency flight of the S-IB stage vehicle. The equations model the six-degree-of-freedom rigid-body dynamics, the gimbaled engines, the aerodynamic and trajectory environment, the first two flexible body bending modes, and the control system feedback networks. An assessment of reliability constitutes a significant part of the study. Therefore, terms were added to model the vehicle behavior following system failures (actuators drifting hardover, loss of engine thrust, nonfunctioning sensors, instruments providing incorrect indications, etc.).

The second-stage simulation was not as complex as the one used for the first-stage study. Structural bending problems are not significant during the S-IVB flight, and aerodynamic terms are negligible for a control system evaluation; therefore, both sets of terms were omitted from the simulation. On the other hand, the first liquid oxygen tank slosh mode is important and was included. A reliability analysis was not conducted for the second stage. The pilot can do nothing to compensate for failures involving the engine since the S-IVB stage has only one engine. Other failures, those associated with the control-system components or instruments, were easily recognized and overcome by the pilot during the first-stage studies. Therefore, it was assumed that there was no need to repeat this class of failures during the second-stage evaluation. A more detailed discussion of reliability considerations is given in a later section.

Manual Control System Simulator

The command pilot's station in the Apollo command module was modeled using a general purpose simulator cab. The cab contained an instrument panel,

a pilot's seat, and a three-axis rotational hand-controller, and the instruments and controller could be connected to the analog computer where the Saturn I simulation equations were solved. The cab arrangement is similar to the one described in reference 3.

Figure 5 is a photograph of the flight simulator instrument panel display used for planning the flight experiment. Centered in the panel is a Flight Director Attitude Indicator incorporating a three-axis ball, cross needles,

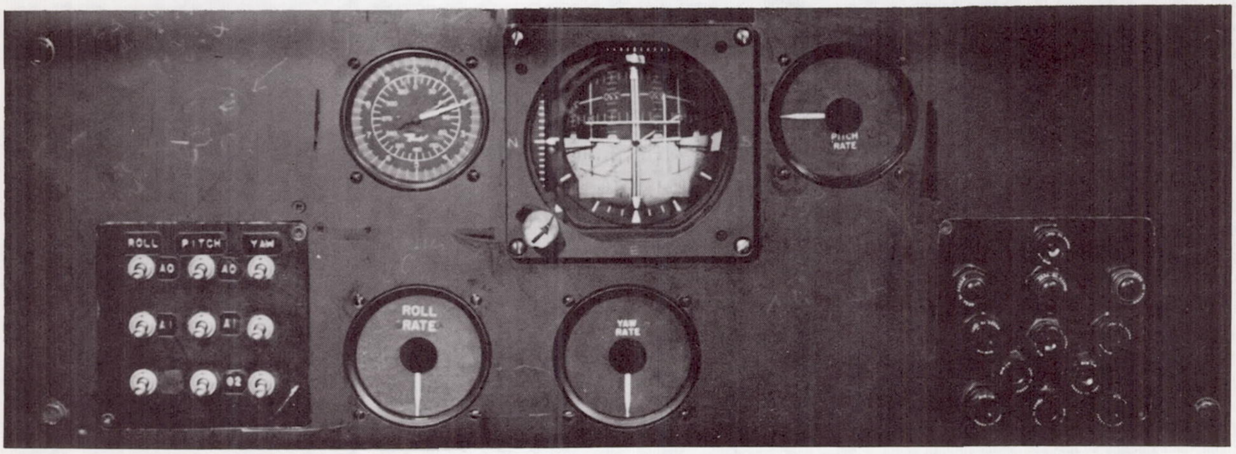

Figure 5.- Simulator display panel.

and side pointers. The total attitude angles (pitch, roll, and yaw) were displayed on the ball. Pitch and yaw attitude error angles were presented on the left side and top pointers, respectively. The cross needles were used to display various quantities. Pitch and yaw normal accelerations were displayed from lift-off through the high dynamic pressure region of flight. For the remainder of the first-stage flight, altitude error and altitude rate error were displayed. During the second-stage flight, the attitude error angles were displayed on the cross needles as well as on the side pointers. The altitude guidance presentation used after high q in the first-stage burn was also evaluated for the second-stage flight.

Three attitude angle rate meters and a clock were grouped around the FDAI: pitch rate on the right, yaw rate below, roll rate on the lower left, and the clock on the left. A set of warning lights was placed in the lower right corner of the panel. These lights indicated which of the Saturn I engines were not thrusting. The light on the right of the horizontal row of lights indicated a launch vehicle inertial platform failure. The remaining lights were not used.

Stability augmentation switches were mounted on the lower left side of the instrument panel. During the simulation studies, the pilot could open a feedback path, that is, do the equivalent of setting a_o, a_1, or g_2 to zero on any axis by moving the appropriate switch to the up position.

A rotational hand-controller, representative of the ones used in the Apollo command module, was mounted on the right arm of the pilot's seat. The

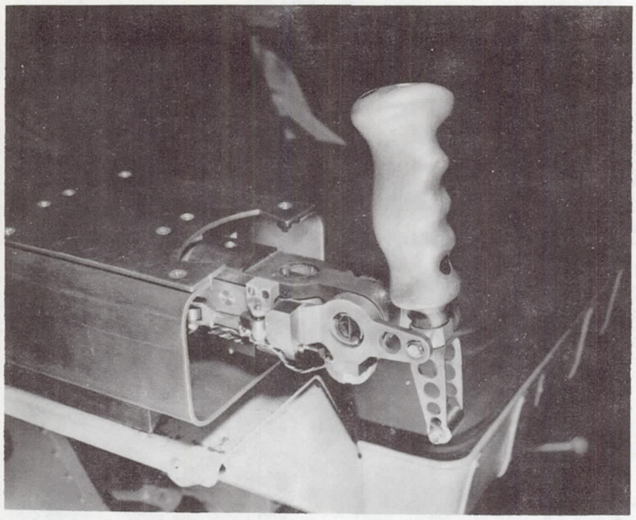

Figure 6.- Three-axis rotational controller.

hand-controller used for the study is shown in figure 6. The handle is pivoted about an axis which permits forward and aft motion in the presence of longitudinal acceleration without introducing inadvertent attitude commands to the simulated vehicle. A roll command is generated when the controller handle is rolled right or left. A pitch command results when the handle is raised up or pushed down. A yaw command is generated when the handle is yawed right or left. Signals to the analog computer are generated by strain gages contained in flexure mounts.

HANDLING QUALITIES STUDIES

The handling qualities studies were undertaken to select a manual control system suitable for the in-flight experiment aboard the Saturn I launch vehicle. The control law adopted for these studies was chosen on the basis of the studies reported in references 1 to 3. This control law is similar to equation (1) and has the following form:

$$\beta_c = a_o N_o \Delta\phi + a_1 N_1 \dot{\phi} + a_p N_p \delta_p \tag{2}$$

where

a_p pilot's rotational controller gain, deg/deg

N_p rotational controller filter

δ_p output signal from pilot's rotational controller, deg

The pilot's rotational controller input is essentially a trim signal introduced only to compensate for unusual flight conditions. So long as the flight proceeds normally, the pilot takes no control action. But in the event of excessive wind shears or a failure in the Saturn I flight control system, the pilot can supply a corrective command signal.

The parameters varied during the handling qualities studies were: pilot's rotational controller gain, a_p, cutoff frequency of the filter on the output of the rotational controller, the attitude error feedback gain, a_o, and the attitude rate feedback gain, a_1. The attitude error and attitude rate

filter parameters were not varied. The handling qualities studies were carried out for both stages and will be discussed separately.

The Cooper rating scale was the performance measure used for the studies. The Cooper rating scale was proposed in 1957 (ref. 5) and is shown here in figure 7. This rating is the pilot's subjective opinion of how well he was able to control the vehicle with respect to some assigned task.

	Adjective rating	Numerical rating	Description	Primary mission accomplished?	Can be landed
Normal operation	Satisfactory	1	Excellent, includes optimum	Yes	Yes
		2	Good, pleasant to fly	Yes	Yes
		3	Satisfactory, but with some mildly unpleasant characteristics	Yes	Yes
Emergency operation	Unsatisfactory	4	Acceptable, but with unpleasant characteristics	Yes	Yes
		5	Unacceptable for normal operation	Doubtful	Yes
		6	Acceptable for emergency condition only*	Doubtful	Yes
No operation	Unacceptable	7	Unacceptable even for emergency condition*	No	Doubtful
		8	Unacceptable - dangerous	No	No
		9	Unacceptable - uncontrollable	No	No
		10	Serious doubt of escape		

*(Failure of a stability augmenter)

Figure 7.- Pilot rating scale.

The following evaluation procedure was used for the handling qualities studies by the two Ames research pilots who participated. Each pilot flew a new situation as many times as he deemed necessary to acquire familiarity. When satisfied that he had developed the appropriate technique, three data runs were taken. At the conclusion of the last run, the pilot gave his opinion as a number from figure 7. In addition, he often added specific comments about unusual features of the run.

Handling Qualities Study for S-IB Stage

Two problems must be considered in defining the S-IB stage manual control experiment. The first problem is high structural loading caused by wind shears and velocities that produce an angle of attack and a corresponding increase in lateral aerodynamic loads on the nose of the vehicle. These loads produce sizable bending moments and also rotate the nose of the launch vehicle away from the relative wind. The control system senses this rotation and sends a command to the engine actuators to reduce angle of attack by rotating the aft end of the vehicle away from the wind. Unfortunately, the thrust component that reduces angle of attack also increases the bending moment. A well-designed control system will use only small actuator deflections to turn the vehicle, with the result that the bending caused by the combination of aerodynamic and control action moments is minimized. Another complication is that the control action that reduces bending moments by heading the nose of the vehicle into the relative wind also causes the vehicle to deviate from the planned flight path. In the Saturn I control system described earlier (eq. (1)), the normal acceleration term in the control law, $g_2 N_2 \ddot{Y}$, is introduced to suppress structural loads and at the same time to keep the vehicle near the planned flight path. For the handling qualities study, the pilot was assumed to be capable of initiating control action to alleviate structural loads. Therefore, the normal acceleration term was eliminated from the control law.

Failure situations constitute the second major problem associated with the first-stage flight. Among the failures that can occur, one of the most

critical to control system operation is a double actuator failure which is introduced in such a way that the nose of the launch vehicle is turned into a relative wind.

A hypothetical combination of events was used as the representative problem for the first-stage handling qualities study. A double actuator failure at 75 seconds after launch was combined with a maximum wind profile that peaks at 70 seconds. The actuators were failed in a direction that made the combined aerodynamic and control moments a maximum. This combination of events constitutes one of the most severe situations that can occur during first-stage flight.

Specific tasks were assigned to the pilot. He was to maintain attitude and minimize bending moments. In addition, control actions were to be smooth so the bending modes would not be excited.

The pilot used variables displayed to him on the Flight Director Attitude Indicator to decide on control actions needed to fly the Saturn I. The primary display variables were the attitude error angle and normal acceleration. The pilot interpreted the normal acceleration indication as a measure of angle of attack. His response to a gradual buildup of normal acceleration was a rotational controller command to turn the launch vehicle into the relative wind without allowing large attitude errors to develop. A rapid increase in normal acceleration and attitude error was a cue that a double actuator failure had occurred. To compensate, the pilot generated controller signals that rotated the three good engines to bias positions to counter the torque produced by the failed actuators. The succeeding rotational controller commands used to continue the flight were perturbations about the bias command.

A preliminary part of the simulator study involved selecting the pilot's controller gain called "control authority." Control authority is the magnitude (expressed in radians) of the command signal to the engine actuators when the pilot's rotational controller is fully deflected. Figure 8 shows how the pilot rating varied with control authority for the pitch and yaw axes. The lowest control authority acceptable (i.e., with a pilot rating of 3) to both pilots was 0.2 radian. Control authority for the roll axis was selected from a similar study and was set at 0.125 radian.

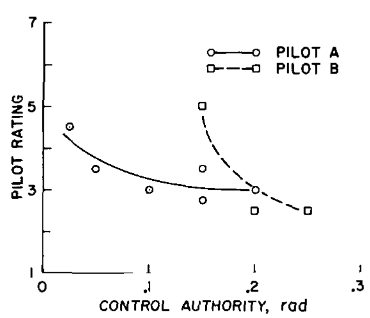

Figure 8.- Pilot rating as a function of control authority for the pitch and yaw axes, S-IB stage.

The output of the pilot's rotational controller was filtered to prevent excitation of the bending modes caused by the high frequency content of pilot commands. The first bending mode of the Saturn I has a natural frequency of approximately 9 rad/sec, which is above the frequencies the pilot generates intentionally in controlling the vehicle. Therefore, a simple first-order low-pass filter was studied for possible use on the output of the pilot's rotational controller. Several filter cutoff

frequencies ranging from 4 to 8 rad/sec were considered. Additional runs were made with the filter omitted. At filter cutoff frequencies below 5 rad/sec, the filter caused a minor degradation in the pilot's ability to control the vehicle. At filter cutoff frequencies above 5 rad/sec as well as with the filter entirely omitted, no significant change in pilot opinion was recorded. The electrical noise from the controller caused some excitation of the second bending mode when the filter was omitted. Since there may be noise associated with the Apollo system, a filter of at least first order should be used on the output of the controller. An adequate filter already available in the Saturn I Flight Control Computer is the attitude error filter, which has a cutoff frequency of from 5.6 to 5.8 rad/sec. Therefore, for a flight experiment, the attitude error and pilot's controller output signals would be summed ahead of the attitude error filter in the Flight Control Computer. This configuration was used for the remainder of the simulation study.

An analytical investigation of the launch vehicle dynamics was conducted using a Laplace transform analysis. This study resulted in plots in terms of the complex frequency operator $s = \sigma + j\omega$ of the characteristic equation root locations for the S-IB stage. The root locations shown in figure 9 are for a representative time of flight, 60 seconds after lift-off, and for the attitude error and attitude rate feedback gain settings planned for a typical Saturn I flight. The roots most strongly affected by changing feedback gains are those associated with the rigid body motion. So long as $0 < a_0 < 2$ and $0.5 < a_1 < 1.8$ the other roots remain close to the locations shown in the figure. When $2 < a_0 < 3$ and $1.8 < a_1 < 3$, the low frequency roots associated with the attitude error and attitude rate networks and the rigid body vary in location with variations of feedback gain settings. The higher frequency roots associated with the attitude rate network and all of the roots associated with the engine, the bending modes, and the trajectory are insensitive to gain settings, a_0 and a_1, of less than 3.

Handling qualities curves for the S-IB stage are shown in figure 10. The heavy lines are profiles of constant pilot opinion using the scale of figure 7. The other lines are profiles of rigid-body natural frequency and damping obtained from the characteristic equation evaluated at difference values of a_0 and a_1. The pilot opinion ratings were related to the dynamics of the

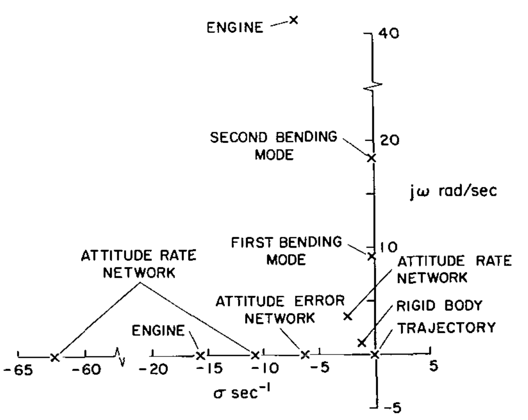

Figure 9.- Roots of S-IB characteristic equation at T = 60 sec.

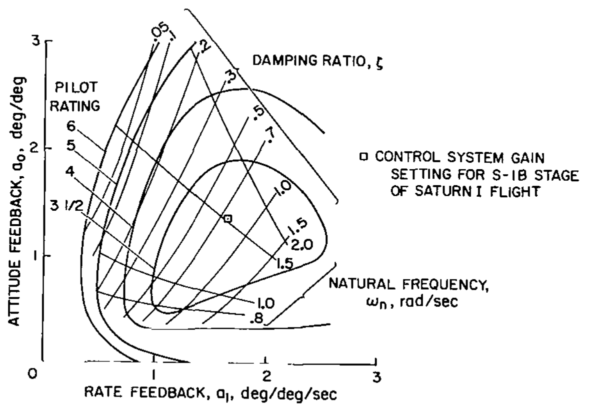

Figure 10.- Pilot rating curves, S-IB stage.

rigid body, which may be considered a second-order dynamical plant. The region inside the 3-1/2 and 4 rating contours indicates that a wide range of values for a_0 and a_1 are acceptable to the pilots. The gain settings used for the Saturn I vehicle first-stage flight are in the most acceptable region of the figure as indicated by the square data symbol. Significantly, the pilots do not regard low values of a_0 unacceptable so long as adequate attitude rate feedback augmentation is provided. When $a_0 < 2$ and a_1 is above 1.5, the rigid-body roots are real and the vehicle behaves primarily as a time constant. With large values of a_1, the response is sluggish and more control authority is desired. The low rigid body damping associated with $a_0 < 1$ and $a_1 < 0.8$ is reflected by increasing values of pilot rating. In the upper right region of the figure, the pilot opinion is influenced by the dynamics associated with both the rigid-body and the low-frequency roots of the attitude rate and attitude error feedback networks. The pilots were essentially attempting to control a system with fourth-order dynamics. Their main comment was that control authority was inadequate.

Handling Qualities Study for S-IVB Stage

The control problem for the second stage is to maintain the launch vehicle at the attitude angle commanded by the iterative guidance system. Since the S-IVB burn takes place above the atmosphere, aerodynamic forces and moments are negligible. The absence of external disturbing forces makes the second-stage burn time ideal for performing the flight experiment maneuvers.

Two maneuvers were used for determining the handling qualities of the S-IVB stage for various a_0 and a_1 gain settings. In the sense of the maximum attitude rate produced, the two maneuvers simulate the load relief and double actuator failure problems of the first stage.

The first maneuver is diagrammed in figure 11 and is called a "step." It results in an average maximum attitude rate near 0.04 rad/sec, which is typical of a double actuator failure. The pilot changes attitude error from 0° to 3° as smoothly and quickly as possible without excessive overshoot. The error is held for 15 seconds, then the vehicle is rotated in the opposite direction.

The other maneuver is the "ramp" shown in figure 12 which results in a maximum rate of 0.01 rad/sec and simulates a load relief problem. The

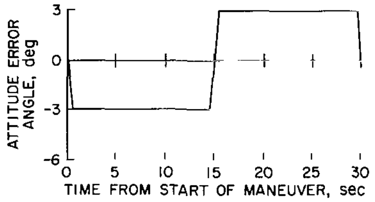

Figure 11.- Maneuver that simulates double actuator failure.

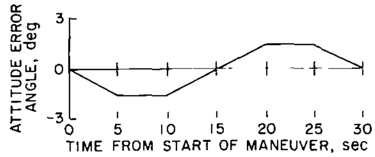

Figure 12.- Maneuver that simulates load relief.

maneuver is started with a 5-second buildup to 1.5° of attitude error. The error is held for 5 seconds, then the ramp direction is reversed until 1.5° of attitude error of the opposite sign is achieved. Five-second intervals were chosen because these are increments that the pilot can easily monitor on a round clock.

The pilot used the attitude error signals displayed on the Flight Director Attitude Indicator cross needles to perform coordinated pitch and yaw maneuvers. The maneuvers were initiated with controller commands which caused the vehicle to pitch down and to yaw to the right. The pilot views this motion as pitch needle motion upward and yaw needle motion to the left. In terms of instrument indications, the intersection of the needles moved along a 45° line marked on the face of the Flight Director Attitude Indicator as the maneuvers were performed.

The two maneuvers were run in sequence; the step was initiated at 3 minutes after lift-off (approximately 30 sec into second-stage flight) and the ramp was initiated at 4 minutes into the flight. The pilots rated the maneuvers separately.

Control authority for the pitch and yaw axes was evaluated for two sets of feedback gains: nominal ($a_o = 0.56$, $a_1 = 0.9$) and reduced ($a_o = 0$, $a_1 = 0.3$). The data from the evaluation are shown in figure 13. Control authority is preferably a constant that does not influence pilot rating as attitude and attitude rate feedback gains are varied. But, it is clear from figure 13 that no single constant will be entirely adequate for all of the range of control authority investigated. The choice of a control authority of 0.07 radian was acceptable for most of the feedback gain settings and was therefore used for the handling qualities study.

Roll control for the S-IVB stage is maintained with on-off reaction jets that are activated whenever the input command exceeds a preset value. As long as the pilot can generate a rotational controller signal exceeding the preset value, it does not matter what control authority is used. To avoid switching control authority at first-stage separation, a single setting (0.125 radian) was selected for both stages.

The characteristic equation roots for the S-IVB stage vehicle dynamics and control system are shown in figure 14. The three sets of roots are for

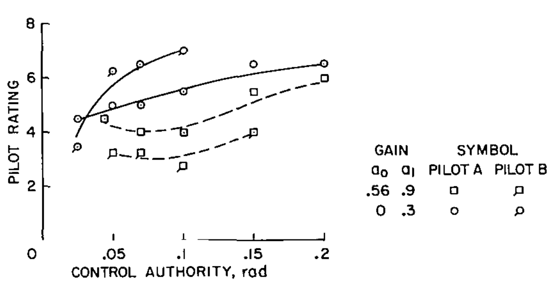

Figure 13.- Pilot rating of control authority for S-IVB.

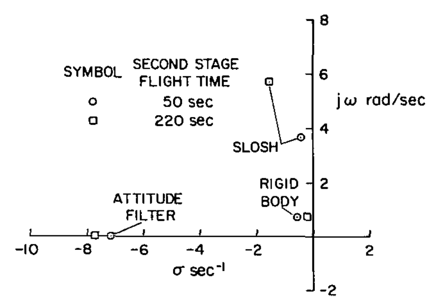

Figure 14.- Roots of S-IVB characteristic equation.

the rigid body dynamics, the liquid oxygen slosh mode, and the attitude error compensation filter. Other roots such as rate networks, actuator dynamics and body bending modes are all at sufficiently high frequencies that they are not significant for the evaluation of vehicle handling qualities. The two sets of root locations shown in figure 14 indicate the dynamics 50 and 220 seconds after initiation of second-stage thrust. During a preliminary assessment, the pilots performed representative maneuvers during two time segments of flight. The pilots could easily accomplish the maneuvers when they were performed early in the second-stage flight. The dynamics in this time segment of flight are represented by the roots labeled 50 seconds in figure 14. The pilot's performance for the maneuvers done during the time segment beyond 200 seconds was poor. This poor performance is related to the gain reduction that takes place at 200 seconds and can be attributed to two factors. As shown by the roots labeled 220 seconds in figure 14, the damping of the rigid body is low. Simulator runs carried out for this region of flight indicate that, in addition to the low rigid body damping, the slosh mode was evident and objectionable to the pilot.

Representative maneuvers should be made early in the second-stage flight because of three factors. First, the vehicle control system dynamics are such that the maneuvers are most easily accomplished prior to the gain reduction which occurs at 200 seconds. Second, the iterative guidance system used during the S-IVB stage burn has ample time to eliminate trajectory errors that might have been introduced by the maneuvers. Third, the dynamics of the vehicle can be altered over wide ranges by varying a_0 and a_1 from the values presently designed into the S-IVB control system. Thus, it is possible to simulate the dynamics at times in excess of 200 seconds by reducing the feedback gains and doing maneuvers earlier in flight.

The upper stage handling qualities plots are shown in figure 15. Equal ratings were usually recorded for step and ramp maneuvers; therefore, only one set of curves is presented. Superimposed on the handling qualities curves are profiles of rigid body frequency and damping. These profiles were determined from the characteristic equation evaluated at 50 seconds into the S-IVB stage flight. The Saturn I control system feedback gains are indicated for 50 and 220 seconds into the second-stage burn by the square and circle data symbols, respectively.

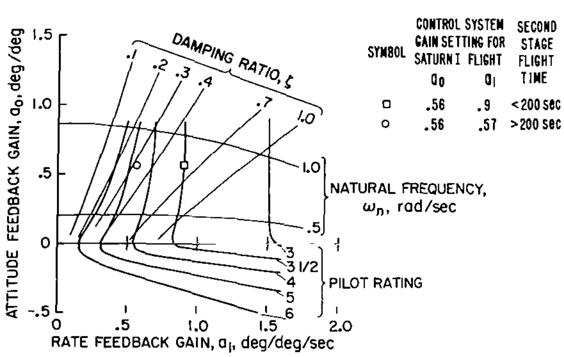

Figure 15.- Pilot rating for simulated actuator failure maneuver.

The a_0 - a_1 plane contains three regions of interest. The first region is bounded on the left by the PR = 4 curve, on the right by the PR = 3 curve, and on the bottom by the a_1 axis. Here the pilot had no difficulty in flying the maneuvers. Damping and control authority were acceptable; the slosh mode was unobservable on the cockpit displays. The second region of interest lies to the left of the PR = 4 line where the

damping is poor. The pilot noticed the slosh mode and considered it objectionable in this region. The last region lies below the a_1 axis. The dynamics in this part of the plane are dominated by a single divergent root and are representative of a statically unstable vehicle. Small negative values of a_o pose no problem to the pilot.

GUIDANCE STUDIES

Techniques for manually guiding the upper stages of Saturn V into an earth orbit were developed during the studies conducted earlier (ref. 2). Some of these techniques were adopted for the Saturn I flight experiment study. The purpose of the present study was therefore more to select suitable guidance display type and scaling rather than to develop guidance techniques.

The displays chosen for the guidance studies utilize the cross needles which are part of the Flight Director Attitude Indicators in the Apollo command module. The variables used for the study were the altitude error and the altitude rate error, which represent the difference between the actual flight path and the nominal pitch program. Two display formats were investigated. For the first of these, a signal which was a linear combination of altitude error and altitude rate error ($a_2\Delta h + a_3\Delta\dot{h}$) was used to drive the horizontal needle of the Flight Director Attitude Indicator. The vertical needle was inactive. The pilot's objective was to null the error signal. The second display, called a vector display, consisted of altitude error (Δh) registered on the vertical needle and the altitude rate error ($\Delta\dot{h}$) registered on the horizontal needle. To null an error using the second format, the pilot first controlled the vehicle so that the needles intersected under a diagonal line marked on the face of the Flight Director Attitude Indicator. Positioning the needles on the diagonal was tantamount to selecting a ratio between Δh and $\Delta\dot{h}$, that is, establishing a control law consisting of a linear combination of altitude error and altitude error rate feedback. Succeeding control action caused the vehicle reference to move to the needle intersection along the diagonal line to null the trajectory error.

During a flight experiment, the primary guidance would be provided by a nominal gravity-turn pitch program during the first-stage burn and by the iterative guidance system during the second-stage burn. Three constraints influenced the study. First, a flight experiment must not interfere with the second-stage iterative guidance system. Second, during the high q portion of the first-stage flight, the vehicle must not fly at angles of attack that would compromise the structure. Third, the attitude rates about all three axes must be nulled prior to staging which occurs at 145 seconds. These three constraints resulted in a decision to start a guidance experiment at 100 seconds into the flight and terminate it at 140 seconds.

A standard procedure was used for the studies. The manual control system used for the study was described earlier by equation (2). It will be recalled that the attitude error and attitude rate feedback paths were retained but the accelerometer feedback was omitted. For the guidance studies, a maximum wind

profile was introduced as a disturbance function. During the simulated flights, the pilot took no corrective action until 100 seconds. This caused the vehicle to deviate from the nominal trajectory so that at 90 seconds the altitude error was 500 m and the altitude rate error was 20 m/sec. For the first 90 seconds of the flight, the Flight Director Attitude Indicator cross needles indicated normal acceleration to provide the pilot with a measure of structural loading. At 90 seconds, the inputs to the needles were changed from normal acceleration to guidance errors. The 10 seconds from 90 to 100 seconds were used by the pilot to assimilate the guidance display information prior to taking over the guidance function. At 100 seconds, the pilot initiated controller commands to reduce first-stage burnout errors subject to the constraint that attitude rates about all three axes were to be nulled at inboard engine cutoff.

The evaluation procedure used by the pilots was the same as that used for the handling qualities studies; that is, the pilot practiced runs until satisfied with his proficiency. Then, he flew three simulated runs for the record. Data recorded consisted of altitude and altitude rate errors measured at 140 seconds and the pilot rating.

Data from the guidance display evaluation are shown in figures 16 and 17. During the studies, the altitude errors were presented so that 1 inch of needle deflection corresponded to 2000 m. Altitude rate errors were presented

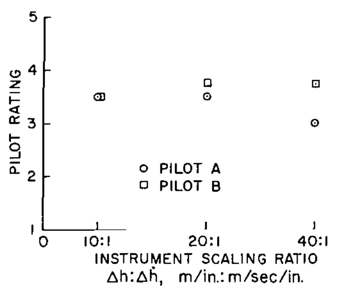

Figure 16.- Pilot rating of $a_2\Delta h + a_3\Delta\dot{h}$ presentation.

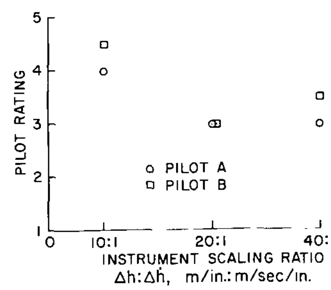

Figure 17.- Pilot rating of Δh and $\Delta\dot{h}$ presentation.

with three different levels of scaling; 200, 100, and 50 m/sec/in. of needle deflection. In figures 16 and 17, the three levels of scaling are shown in terms of ratios identified in the following table.

Ratio	Altitude error, m/in.	Altitude rate error, m/sec/in.
10:1	2000	200
20:1	2000	100
40:1	2000	50

There was little variation in the pilot rating data for either display. Figure 16 indicates that, for the first display, the linear combination of altitude error and altitude rate error, the pilots considered all three ratios to be equally effective. The data for the second display shown in figure 17

indicate that ratios of 20:1 and 40:1 were favored over 10:1. No preferences between the two systems were noted during the guidance studies. However, later during the reliability assessment, which will be discussed in the next section, one of the pilots commented that the vector display was sometimes confusing. It did not seem natural to him to have an altitude indication displayed on a vertical needle. There was little variation in the end-point dispersion data for either display. The average trajectory error at inboard engine cutoff was 100 m and 7 m/sec. This compares to 800 m and 6 m/sec if the pilot made no guidance correction.

RELIABILITY EVALUATION

The purpose of the reliability evaluation was to show that mission reliability will not be impaired by the flight experiment modifications and procedures. Since an extensive analysis had already been completed for the Saturn V (ref. 3), only an abbreviated reliability analysis of the Saturn I was necessary to account for the differences between the two vehicles. The evaluation procedure given in reference 3 was adopted for the Saturn I study. Briefly, this procedure is:

1. Define the system.

2. Define major failure modes.

3. Simulate system and failure modes.

4. Define pilot procedures.

5. Conduct simulation with random failures.

6. Calculate probability of mission failures.

The reliability study is discussed in the order listed above.

The reliability analysis was conducted for only the first stage. The reasons for limiting the study to the first stage are summarized below.

1. Structural loading associated with first-stage flight is a major concern in the design of the Saturn I control system.

2. The peak dynamic pressure and the wind disturbances occur during the first-stage burn.

3. The four gimbaled and controllable engines on the S-IB stage make it possible to compensate for most failures associated with a single engine.

4. The results of the first-stage reliability study involving failures such as sensors or instruments are assumed to be applicable to the upper stage.

For the reliability analysis, the control system feedback gains were set to $a_0 = 0.2$, $a_1 = 1.65$, and $g_2 = 0$. This gain configuration represents the minimum booster augmentation that would be proposed for a flight experiment.

Failure Modes

An assessment was made of the launch vehicle failure modes (as opposed to component failures) that could occur to interfere with the controllability of the Saturn I. The most critical of the failures and the corresponding probabilities of occurrence are listed in table II. Four of the failure modes listed were excluded from the simulation studies. Three of these (items 14-16) were omitted because the probability of occurrence is negligibly small. The remaining failure, that is, one actuator null, was dropped because it represents a less severe problem to the pilot than the fully deflected actuator considered.

Types of failures were combined with the following flight conditions to produce a test series of 82 simulated flights. The first-stage flight was divided into three time segments: before, during, and after high q. Two wind conditions, the 95- and 50-percent profiles, and the two wind directions, northwest and southwest, were considered. A single actuator could fail toward or away from the wind. A thrust or double actuator failure could occur to turn the launch vehicle into, out of, or perpendicular to the wind. There was no direction associated with the other failure modes (e.g., a rate gyro failure). An individual situation consisted of a combination of the region of flight when the failure occurred, the type of failure, and the direction of failure relative to the wind if more than one direction was possible. All situations were encountered at least once during the simulation series. Extra situations were included for those failures which have a high probability of occurrence. The number of simulated flights for each type of failure is listed in the last column of table II.

During the Saturn V simulation studies (ref. 3), instrument display failures caused no mission failures. To verify this conclusion for the Saturn I, a limited number of instrument and accelerometer failures were introduced - items 8 through 13 in table II.

Pilot Procedures

In the event of a system failure, the pilot first identified the problem. He then used the rotational hand controller to maintain the vehicle at or near nominal values. Different control techniques were needed for the various categories of failures. For hardware malfunctions in the launch vehicle control system (loss of inertial attitude, attitude rate, attitude error angle feedback, etc.), the pilot used information from sensors located in the spacecraft to stabilize and control the vehicle attitude. For example, a launch vehicle attitude rate loop malfunction (failure 6 in table II) caused the vehicle motions to become dynamically unstable. The pilot compensated for this failure by using the displayed rate information, which is sensed by

gyros located in the Apollo command module, to stabilize the vehicle motions. In the case of an engine actuator failure or loss of thrust, asymmetrical rotational moments were developed on the vehicle. To correct these situations, the pilot acted as an integration type element to inject trimming or bias commands which nulled the rotational moments caused by the failure. In the case of a single display failure, the instrument indications in the command module were sufficiently redundant that, by cross-checking, the pilot was able to detect which instrument had failed. He could continue to fly the vehicle using the remaining valid indications.

On all flights in the series, the pilot used the normal acceleration indications up to 90 seconds to reduce the vehicle structural loading. At 90 seconds, the Flight Director Attitude Indicator cross needles were switched to indicate altitude error and altitude rate error. The pilot used these guidance error indications to reduce miss distances at first-stage burnout.

Simulation Procedure

To compare manual and Saturn I fully automatic control system performance during failure, three sets of runs, each consisting of the 82 situations, were conducted. Two of these were with pilots flying the simulated launch vehicle using a manual control system, the third set was done with the Saturn I automatic control system.

The pilots were exposed to failure situations in a randomly ordered sequence. Before each simulated flight, the pilot was briefed on wind direction and magnitude. (This wind information is available to the astronauts prior to an actual flight.)

For all situations except an engine out, the criterion for a successful mission is that the maximum normalized bending moment experienced during flight must not exceed unity. This normalized bending moment is expressed as the ratio of the maximum bending moment during the flight to the breakup bending moment. So long as breakup did not occur, the vehicle was flown as near to the first-stage burnout aim point as the pilot could manage. Pilot rating, maximum bending moment, maximum attitude rate, and the burnout miss distance from the nominal trajectory were all used to judge how well the mission could be flown. The criterion for a successful engine-out flight was different. In this case, the mission was considered successful if the launch vehicle could be flown through high q without breaking up. Unless the engine failure occurred very late in flight, it would not be possible to stay near the nominal trajectory.

Results of the Reliability Evaluation

The results of the reliability analysis phase of the simulation studies are presented in three parts: the manual and Saturn I control system performance for each failure; the impact of this performance on mission reliability; and the evaluation of trajectory dispersions.

Manual and Saturn I control system performance.- Figure 18 shows the maximum attitude rate, pilot rating, and maximum normalized bending moment for each type of failure listed in table II. The maximum and average values of each parameter are shown for the Saturn I control system and for both pilots. The first three columns show behavior in the presence of the 95 percent wind; the last three columns are for the 50 percent wind.

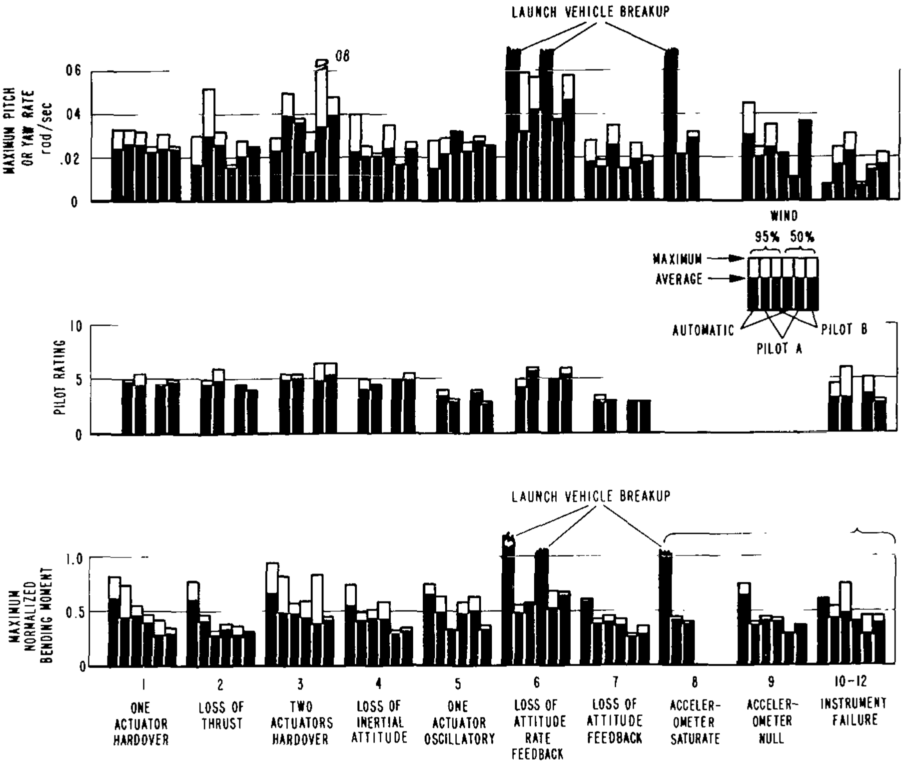

Figure 18.- Maximum pitch or yaw attitude rate, pilot rating, and maximum bending moment following various failures.

The performance of the Saturn I automatic flight control system is indicated in the first (95-percent wind) and fourth (50-percent wind) columns of figure 18 for each failure type. The maximum normalized bending moment exceeded unity (vehicle breakup) during some of the loss of attitude rate feedback and all of the accelerometer-saturate situations. A third critical failure was two actuators hardover for which the bending moment reached 0.95 for one situation.

The pilot's performance is shown in the second and third columns (95-percent wind) and the fifth and sixth columns (50-percent wind) of the figure for each failure situation. None of the manually controlled missions were terminated by launch vehicle breakup. The pilots considered the two actuators hardover and loss of attitude rate feedback to be the most difficult to fly. Their maximum rating, 6-1/2, was given for the double-actuator failure. Loss-of-attitude-rate-feedback failures were considered uniformly hard to fly; maximum pilot rating was 6 and the average was 5-1/2. The other failures were not troublesome.

Certain of the reliability results from figure 18 are related to the handling quality studies described earlier. Recall that the representative problem used for the earlier evaluation included a double-actuator failure. The average pilot rating for the double actuator failures (fig. 18) is 5. This compares to a pilot rating of 4-1/2 in the first-stage handling qualities study. For the handling qualities study, the pilots knew what the failure was and when it would occur. But, for the reliability analysis, there was an additional element of surprise; the pilot did not know ahead of time when the failure would occur or what it would be. It is evident that the element of surprise is not critical since the results of the two study phases are in good agreement.

The noncritical (instrument and accelerometer) failure flights are representative of normal missions. During these flights, the only disturbance applied to the vehicle was the wind profile. Average pilot ratings near 3 were given for these flights.

Another observation relates to the step maneuver used for the S-IVB stage handling qualities studies. In figure 18, the average value of the maximum attitude rates lies just below 0.4 rad/sec. The step maneuver, which also results in rates near 0.4 rad/sec, is therefore considered to provide the pilots with a problem approximately as demanding as a double-actuator failure.

Mission reliability.- Criticality numbers calculated using the equations given in appendix A on data reduction procedures are shown in table III. A low probability of failure is associated with each of the types of failure for which the automatic control system could not complete the mission. Therefore, the total mission criticality numbers were small - 14×10^{-6} for a 95-percent wind and 7×10^{-6} for a 50-percent wind.

No failures were recorded for the manual control system; hence, the total mission criticality number is zero.

The data presented in figure 18 show that the pilots were always able to prevent aborted missions due to excessive structural loading. This is a better record of performance than was obtained for the Saturn V reliability study (ref. 3) where some situations resulted in launch vehicle breakup. The superior Saturn I record is attributable to at least two factors: structural strength of the Saturn I vehicle and level of pilot training. Because of the sample sizes of 82 cases, there may be situations other than those covered in the study that the pilots will be unable to fly. On the other hand, the Saturn I is shorter, stronger, and therefore easier to fly than the Saturn V. Because the pilots participated extensively in the Saturn V study and therefore had a backlog of experience, they had only a short period of training on Saturn I failures (eight hours in addition to the time devoted to the handling qualities study). With more training, their ability to fly the missions would certainly improve. It is apparent that a Saturn I manual control system can provide acceptable reliability for the in-flight experiment.

Trajectory dispersions.- After high q, the pilot was mainly concerned with reducing first-stage burnout dispersions. Any trajectory errors introduced in the flight-path (pitch) plane due to wind disturbances were nulled

during the final 40 seconds of flight with the aid of the guidance displays. Performance of the automatic control system and of each of the pilots is shown in figure 19, where the averages of the magnitudes of errors in altitude, altitude rate, lateral position, and lateral velocity are plotted. Representative dispersions are shown in the figure for all nonabort, single-actuator, double-actuator, and all instrument or accelerometer failures.

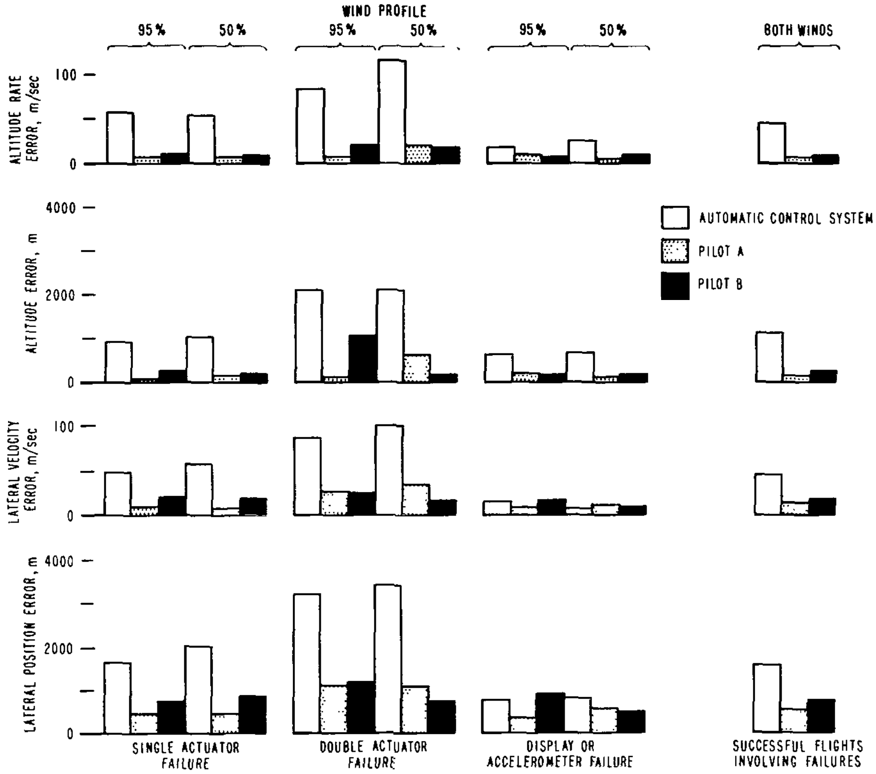

Figure 19.- Average dispersion errors at first stage burnout for various failures.

The pilot's performance is notably better than that of the Saturn I control system. This is understandable for the pitch plane cases since the Saturn I control system nulled only attitude and attitude rate errors; no feedback paths existed in the control system for nulling trajectory errors. On the other hand, the pilots had attitude, attitude rate, altitude, and altitude rate displays so they could manually provide a guidance feedback path. Their success in controlling the altitude dispersions is demonstrated in the figure.

The pilots also performed well in nulling lateral dispersions, especially for engine actuator failures where a large attitude torque bias exists on the vehicle. By acting as an integration type element (biasing the pilot's hand controller) the pilot was able to maintain close attitude control. Even though lateral dispersion data were not displayed to the pilot, the lateral dispersions for the piloted flights were considerably smaller than for the flights with automatic control.

CONCLUSIONS

The following conclusions were drawn from the simulation study of the Saturn I manual control system.

1. The display variables required by the pilot are related to the mission task. For the load relief task the pilot needs indications of attitude error and structural loading. Structural loading information can be in the form of normal acceleration measured at the Instrument Unit of the Saturn I or it can be angle of attack sensed at the nose of the launch vehicle. Variables which are important for the guidance experiments are altitude error, altitude rate error, and attitude error. Attitude error, attitude rate error, normal acceleration (angle of attack), altitude error, and altitude rate error displays are all needed if a failure occurs since they are aids for continuing the flight if continuation is possible. Attitude rate error is primarily an indication of impending catastrophic failure and is of secondary interest for flying the launch vehicle.

2. There is considerable latitude in the choice of feedback gains for the manual control system. The acceleration feedback loop of the Saturn I control system can be omitted when the vehicle is under manual control. Attitude error feedback (a_0) can be significantly reduced. Attitude rate feedback (a_1) must be maintained at or near the MSFC design level. So long as attitude rate feedback compensation is provided, the pilots regard the handling qualities as adequate.

3. Attitude maneuvers can be conducted in the early portion of the S-IVB stage flight without causing significant trajectory perturbations.

4. The type of display and the scaling over the range studied do not influence the pilot's ability to complete the guidance task provided the display indicates altitude error and altitude rate error and provided guidance is the only task assigned to the pilot.

5. The simulation studies indicate that the addition of a manual control system to the Saturn I will improve the reliability of the launch vehicle.

Ames Research Center
National Aeronautics and Space Administration
Moffett Field, Calif., 94035, Nov. 22, 1968
125-19-01-39-00-21

APPENDIX A

DATA REDUCTION PROCEDURE FOR THE RELIABILITY ANALYSIS

The basic laws of probability needed to compute the reliability of the Saturn I manually controlled booster are written in concise form in reference 6. These laws are expressed in terms of two events, A and B, that have occurred during a hypothetical experiment. The probability of A occurring is written $P(A)$ which is the number of times that A occurred divided by the total number of trials. The probability associated with event B is $P(B)$. The event that both A and B occur is written AB. The associated probability is $P(AB)$. The probability that either A or B or both occur is written $P(A + B)$. The following relation can be deduced:

$$P(A + B) = P(A) + P(B) - P(AB) \tag{A1}$$

If A and B are mutually exclusive, that is, A and B do not occur together, then $P(AB)$ in equation (A1) is zero and $P(A + B)$ is the sum of $P(A)$ and $P(B)$.

The probability that A occurs, given that B has occurred, is called a "conditional probability" (also "effectivity number") and is written as $P(A/B)$. It can be shown that

$$P(AB) = P(B)P(A/B) \tag{A2}$$

For the specific launch vehicle experiment - the introduction of random failures as the pilot flies the booster simulation - the following definitions and assumptions are established. Denote a launch vehicle failure event by the symbol F and associated probability of failure by $P(F)$. A type of failure - one of those listed in table II - is given the symbol C_i and associated probability of occurrence, $P(C_i)$. It is assumed that the $P(C_i)$ can be predicted by a Poisson process. That is, once the components are installed, operated long enough to eliminate manufacturing failures, and tested, the succeeding probability of failure is proportional to the time the component is operated. Since the $P(C_i)$ for all the types of failures are small numbers, the probability that more than one failure will occur during a flight is considered negligibly small. For present purposes, the types of failures are essentially mutually exclusive.

The probability that the mission will fail if one of the types of failures occurs is called the "criticality number," and is denoted by $P(FC_i)$. From equation (A2),

$$P(FC_i) = P(C_i)P(F/C_i) \tag{A3}$$

The probability that the vehicle will fail, given that a type of failure has occurred, $P(F/C_i)$, is obtained from the simulation experiment. For example, if a pilot flew three missions with a given type of failure and could complete only two of the missions, $P(F/C_i)$ would be 1/3.

Since an unequal number of situations were flown in each of the three time segments, two properties - independent events and reliability defined by the Poisson process - were applied to arrive at the component criticality number. First, the criticality numbers for each of the time segments (0-60 sec, 60-100 sec, and 100-140 sec) are mutually exclusive and can therefore be directly summed:

$$P(FC_i) = P(FC_i)_{0-60} + P(FC_i)_{60-100} + P(FC_i)_{100-140} \tag{A4}$$

The probability of a component failure for a Poisson process is written as

$$P(C_i) = \beta \Delta t \tag{A5}$$

where β is the number of failures per million flights per unit of time. The time interval, Δt, is the length of the flight, 140 seconds for the S-IB stage. To determine the probability of a component failure in each of the time segments, $P(C_i)$ is multiplied by the ratio of time in the segment divided by the length of the flight:

$$\left.\begin{aligned} P(C_i)_{0-60} &= P(C_i)\,\frac{60}{140} \\ &\text{and} \\ P(C_i)_{60-100} = P(C_i)_{100-140} &= P(C_i)\,\frac{40}{140} \end{aligned}\right\} \tag{A6}$$

Taking equations (A3) and (A6) into account, equation (A4) becomes

$$P(FC_i) = P(C_i)\left\{\frac{60}{140}\,P\left(\frac{F}{C_i}\right)_{0-60} + \frac{40}{140}\left[P\left(\frac{F}{C_i}\right)_{60-100} + P\left(\frac{F}{C_i}\right)_{100-140}\right]\right\} \tag{A7}$$

Finally, since each failure is treated as mutually exclusive, the total criticality number is the sum of the criticality numbers for each type of failure.

An example pertaining to the automatically controlled S-IB stage follows. With a 95-percent wind, a total of three loss-of-attitude-rate-feedback simulated flights were flown, one in each time segment. From table II, $P(C_6)$ is 17×10^{-6}. When the booster was under automatic control, the booster broke up following failures which occurred before and during high q. Therefore, $P(C_6)_{0-60} = 1$, $P(F/C_6)_{60-100} = 1$, and $P(F/C_6)_{100-140} = 0$. It follows that the criticality number for item 6 is

$$P(FC_6) = (17\times10^{-6})\left[\frac{60}{140}\,(1) + \frac{40}{140}\,(1) + \frac{40}{140}\,(0)\right] = 12\times10^{-6} \tag{A8}$$

The criticality numbers for all of the failure types are listed in table III.

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TABLE I.- EVENT SEQUENCE FOR SATURN I

Flight time, sec		Event
S-IB	S-IVB	
0		Lift-off
60		Initiation of high q
70		Maximum q for simulation
100 110.0 120.0		Attitude and rate feedback gain changes
141.4		Inboard engine cutoff
144.40		Outboard engine cutoff
145.28		Physical separation
146.6	-3.3	S-IVB engine start command
149.9	.0	S-IVB 90-percent thrust
157.2	7.3	Ullage jettison
184.4	34.5	Launch escape tower jettison
349.9	200.0	Attitude rate feedback gain reduction
476.9	327.0	Mixture ratio change
600.48	440.58	S-IVB engine cutoff command

TABLE II.- FAILURE MODES

Failure number	Type of failure	Probability of failure × 10^6	Number of simulated flights
1	One actuator hardover	11,161	18
2	Loss of thrust	5,000 (Est.)	13
3	Two actuators hardover	10,200	13
4	Loss of inertial attitude	2,035	5
5	One actuator oscillatory	156	5
6	Loss of attitude rate feedback	17	5
7	Loss of attitude feedback	17	5
8	Accelerometer saturate	6	2
9	Accelerometer null	95	4
10	Attitude error display failure	*	4
11	Attitude rate display failure	*	3
12	Accelerometer display failure	*	3
13	Inertial attitude display failure	*	2
14	Saturation of inertial attitude	**	0
15	Saturation of inertial attitude error	3	0
16	Saturation of attitude rate	<1	0
17	One actuator null	28	0
	Total number of simulated flights		82

*No information available.

**The inertial platform is buffered by the redundant Launch Vehicle Data Adapter. Reasonableness checks are made and saturated failures are not permitted.

TABLE III.- MISSION CRITICALITY ESTIMATE

Failure number	Type of failure	Probability of failure $\times 10^6$	Criticality $\times 10^6$			
			95-percent wind		50-percent wind	
			Auto	Piloted	Auto	Piloted
1	One actuator hardover	11,161	0	0	0	0
2	Loss of thrust	5,000 (Est.)	0	0	0	0
3	Two actuators hardover	10,200	0	0	0	0
4	Loss of inertial attitude	2,035	0	0	0	0
5	One actuator oscillatory	156	0	0	0	0
6	Loss of attitude rate feedback	17	12.2	0	7.3	0
7	Loss of attitude feedback	17	0	0	0	0
8	Accelerometer saturate	6	1.7	0	0	0
9	Accelerometer null	95	0	0	0	0
10	Attitude error display failure	*	-	0	-	0
11	Attitude rate display failure	*	-	0	-	0
12	Accelerometer display failure	*	-	0	-	0
13	Inertial attitude display failure	*	-	0	-	0
	Total mission criticality		14	0	7	0

*Not applicable.

Definition of criticality: Probability that the mission will fail if one of the types of failure occurs.

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